Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	MET: Prelaunch	Problem	FIAR	IFA STS-40-V-01	GN&C
None	GMT: Prelaunch		SPR 40RF01	UA	Manager:
			IPR None	PR	
					Engineer:

Title: Inertial Measurement Unit 2 failed Pre-Flight Calibration (ORB)

Summary: DISCUSSION: During prelaunch calibration, the inertial measurement unit (IMU) 2 (ser. no. 23 in slot 2) experienced out-of-specification X and Y accelerometer biases and scale factor shifts.

CONCLUSION: Postflight testing at JSC has isolated this problem to a defective computer interface card (CIC) in IMU 2. The failed CIC resulted in one axis sensing an input rate at the same time another axis was transitioning through a zero rate value. This caused a scale factor error that resulted in a failed calibration. CORRECTIVE_ACTION: The defective CIU will be removed and replaced and returned to the vendor for evaluation. The specific mechanism internal to the CIC that resulted in this failure is unknown; however, this failure is a very unique occurrence and does not represent a generic CIC (or IMU) problem. Once a good CIC is installed in IMU 2, it will then be checked out and returned to the fleet. EFFECTS_ON_SUBSEQUENT_MISSIONS: Should this failure occur during flight, it is detectable when data exceed redundancy management (RM) limits. The failed IMU would then be deselected, leaving 2 IMU's available. According to current flight rules, this would also result in an early mission termination. This problem is also detectable during the prelaunch IMU calibration.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	MET: 000:02:30	Problem	FIAR	IFA STS-40-V-02	STR,TPS
MMACS-01	GMT: 156:15:55		SPR 40RF03, 40RF02	UA	Manager:
			IPR None	PR	
					Engineer:

Title: Aft Bulkhead/Payload Bay Door Interface Damage. (ORB)

Summary: DISCUSSION: Video downlinked during the payload bay door opening showed that several thermal blankets on the aft (Xo-1307) bulkhead had become partially unfastened during ascent. Payload bay cameras B and C were used to inspect the loose blankets on the aft bulkhead. Further downlinked video was requested that verified that the payload bay door environmental seal had become detached from the holding track between Yo=-10.7 to Yo=-41 at Xo=1307. The effects of the debonded seal and loose multilayer insulation (MLI) blankets on payload bay door closure and latching, as well as the thermal effects of the loose blankets on nearby hardware, were evaluated.

The JSC Flight Science Support Office used the video from payload bay cameras B and C, video taken by the crew with the onboard camcorder, and Orbiter and Spacelab engineering design data to perform a triangulation analysis to better define the location of the seal in Orbiter coordinates, and to estimate the length of each of the two seal pieces that had come out of the track and were extending into the payload bay. The analysis showed that approximately 3 ft. of the seal material had become debonded and that the location of the tip of the longer piece was extending forward approximately 5 inches and downward approximately 3 inches; the shorter piece was extending approximately 2 inches forward. The positions of the thermal blankets and locations of the remaining fastened snaps were also evaluated. The study showed that the potential existed for one of the thermal blankets (V070-363643-026) and the loose environmental seal pieces to interfere with payload bay door closure. The analysis also yielded information about the way the seal pieces moved in temperature changes from exposure to the sun and when in darkness. Several possible repair procedures were considered. Consideration was given to cutting off the loose seal pieces or reinserting the loose seal pieces into the retainer. A section of seal material was shipped from KSC to JSC to be used in developing possible in-flight maintenance (IFM) procedures. A team traveled to KSC to perform tests using vehicle OV-103 to aid in evaluating possible extravehicular activity (EVA) and IFM procedures. Actual hardware was used to determine the force required to reinsert the seal into the track. The testing results indicated that door closure confidence was high using normal payload bay door closure procedures and that the proposed EVA/IFM activities could be performed, if needed. Rockwell-Downey analyzed the venting and structural effects of the missing seal, as well as the thermal effects on the orbital maneuvering system (OMS) high-point bleed line and auxiliary power unit (APU) 2 which were located near the loose thermal blankets. The analysis showed that even with the missing seal material, the aft bulkhead temperatures would remain within the certification range for the structure, and no potential thermal concerns existed for the as-planned STS-40 attitude timeline. Potential concerns for door closure and entry heating and venting pressure was evaluated. The payload bay door/bulkhead gap sensitivity to Orbiter longeron and bottom skin temperature was analyzed, and the effects of on-orbit temperatures on panel lug-to-aft-bulkhead gap clearances was documented. Further analysis determined that with a 5-ft. piece of the seal missing, the thermal and pressure conditions would be acceptable, and that the payload bay door drive mechanism provides more than enough force to overcome seal resistance. A structural stress analysis was then performed to evaluate the effects of forced closure on the door structure. The analysis determined that if the left door was displaced, the right door corner would deflect/break and permit the latches to fully engage. Further analysis was performed to evaluate alternate end-of-mission (EOM) attitudes to provide both seal warming as well as meet environmental control and life support system (ECLSS) radiator cold-soak requirements. A short nose-sun attitude was chose to warm the seal. A decision was made that upon completion of the science experiments, the seal would be thermally conditioned and manual door closure would be used with EVA as a contingency. The payload bay door seal was thermally conditions by going nose-to-sun 1.8-degree pitch-down attitude for a 30-minute period prior to port door closure. The port payload bay door was closed at 165:11:20:23 G.m.t. using a manual procedure to allow another 30-minutes before closing the starboard door. The door was closed using single motor drive (power system 2, motor 1) and all six ready-to-latch indications were received (three for the forward bulkhead, three for the aft bulkhead); the forward door closed indication talkback was received first and this inhibits motor number 1, stopping it before the aft door closed indication was received; this was expected, and the aft door closed indication was picked up when power system 1 was enabled for dual motor drive for the forward and aft bulkhead latches. The starboard door closed nominally at 165:12:08:53 G.m.t. after completing the manual operation. The doors opened nominally postflight. Chit J3596A was approved to perform postflight inspections at Edwards Air Force Base and Chit J3595A was approved to perform further inspections at KSC. One piece of the seal was found lodged where the guide roller had pressed it down into the hook mechanism. Chit J3595A will also collect data to verify that the rigging is still within specifications and that no structural deformation existed as a result of compressing the seal in the bottom of the hook. Chit J3419A is in the approval cycle and requests that payload bay door bulkhead roller/door gap measurements be taken on all vehicles. The cause of the environmental seal and thermal blanket damage is currently not known. Based on data from previous flights where blanket snaps unfastened (STS-27 and -29), a piece of debonded seal material was found behind the thermal blankets (STS-41), undefined airflow past the payload bay door/aft bulkhead interface could be the cause of the problem. Chit J3450B was approved to inspect the OV-102 environmental seal to insure its integrity prior to STS-35 and to inspect OV-104 prior to STS-38; Chit J3501 was approved to investigate possible door interference. The environmental seal failed at a material review (MR) splice. Chit J3502R1 was approved to inspect vehicles OV-102 and -104 and record all seal installation discrepancies, after the STS-41 seal failure investigation revealed that some of the seal joints had not been installed per drawing. Further analysis/testing will be performed to understand the sensitivity and margin associated with as built tolerances, structural deflections and age of seal material. CONCLUSION: A missing payload bay door environmental seal is not a safety-of-flight issue. The potential exists for interference with payload bay door closure, and for damage to the thermal blankets located on the aft bulkhead. The cause of the environmental seal and thermal blanket damage is currently not known. Undefined airflow past the payload bay door/aft bulkhead interface is potentially the cause, a contributing factor, or a symptom of the problem. Further analysis/testing will be performed to understand the sensitivity and margin associated with the as-built tolerances, structural deflections and age of the seal material. CORRECTIVE_ACTION: Supplemental flight data is desired; as yet, a cost effective implementation has not been defined. EFFECTS_ON_SUBSEQUENT_MISSIONS: Payload bay door environmental seal damage is not a safety-of-flight issue. Possible mission impacts include real-time modifications to the attitude timeline, shortened duration, and EVA/IFM procedures.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:** 000:21:22	Problem	**FIAR**	**IFA** STS-40-V-03	STR
MMACS-02	**GMT:** 157:10:47		**SPR** 40RF04	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: LiOH Door Aft Port Latch Closure Interference (ORB)

Summary: DISCUSSION: On flight day 2, the crew reported that the aft port latch on the LiOH stowage door was stuck closed. In-flight maintenance (IFM) tools were used to pry the latch open, and the latch access was secured with tape. Analysis showed that no structural concerns existed for either entry or a nominal landing with the latch open, but crash-load requirements would have been violated. The crew was able to close the latch prior to entry using onboard tools. The latch operated freely postflight, however, further analysis showed that the total tolerance build-up on this latch and fitting can result in a structural interference at 1-g (gravity acceleration). To ensure proper latch insertion on future flights, engineering is evaluating the latch fittings to verify that the maximum tolerance specified by the design drawings exists on the latch-fitting openings. If not, the openings will be reamed to the maximum tolerance. Should the openings already have the maximum tolerance, an Engineering Order (EO) will be opened to increase the latch fitting opening clearances beyond the current maximum tolerance. This would require a new modified configuration for the fleet. There have been no problems with these latches on the other vehicles that have required in-flight maintenance; however, a tight fitting latch that was difficult to operate was mentioned by the crew on an earlier flight of OV-103, but was not identified as a problem.

CONCLUSION: The in-flight LiOH door latch interference resulted from a cumulative tolerance build-up.   CORRECTIVE_ACTION: Engineering is evaluating the latch fittings to verify that the maximum tolerance specified by the design drawings exists on the latch fitting openings. If not, the openings will be reamed to the maximum tolerance. If the openings already have the maximum tolerance, an EO will be opened to increase the latch fitting opening clearance beyond the current maximum tolerance (requires a new modified configuration for the fleet).   EFFECTS_ON_SUBSEQUENT_MISSIONS: Possible IFM procedures.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:** 001:06:15	Problem	**FIAR**	**IFA** STS-40-V-04	OMS
PROP-01	**GMT:** 157:19:40		**SPR** 40RF05	**UA**	**Manager:**
			IPR 50V-0004	**PR**	
					Engineer:

Title: OMS Crossfeed Line Heater A failed to Cycle On (ORB)

Summary: DISCUSSION: After having cycled twice within the normal temperature operating range while operating on the orbital maneuvering system (OMS) crossfeed heater A system, the OMS oxidizer crossfeed center line temperature (V43T6242A) dropped below its prior minimum point of approximately 62 ?F. The temperature had decreased to 55 ?F when the crew switched to the OMS crossfeed heater B system at 157:20:01 G.m.t. (1/06:36 MET). The temperature returned to its normal operating range at this time and remained nominal for the remainder of the mission.

Postflight troubleshooting revealed a problem in the heater thermostat switch (S1052). At the time of this report, however, removal of the thermostat was being scheduled to be accomplished at KSC after completion of the OV-102 Orbiter Maintenance Down Period at Palmdale.   CONCLUSION: The most probable cause of the apparent failure of the OMS oxidizer crossfeed center line heater to activate was a failure in the heater thermostat switch.   CORRECTIVE_ACTION: The switch will be R&R'ed.   EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	**Time**	**Classification**	**Documentation**		**Subsystem**
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-40-V-05	GFE

MMACS-05	**GMT:** 158:23:10		**SPR** None	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: Video Interface Unit-C Power Cable Anomaly (GFE)

Summary: DISCUSSION: During camcorder operations, the camcorder did not operate off the video interface unit (VIU)-C power but operated with battery power. An in-flight maintenance (IFM) procedure was performed and revealed an open circuit in the adapter cable power lines. The video lines were verified to be operating nominally.

CONCLUSION: Troubleshooting postflight revealed two causes of this problem. A bad connection on the cable was one cause. The connector potting did not hold to the Teflon cable well enough to provide the necessary strain relief, failing to prevent a broken wire in the cable. A second possible cause was the VIU power DC/DC converter being adjusted slightly undervoltage for the new cable. The camcorder requires a high power-on surge current, something batteries are good at providing, but DC/DC converters are not. This coupled with the new cable having smaller gauge wire than the original cable, created a voltage drop across the cable that was recognized by the camcorder shutting itself down. The VIU undervoltage problem may have caused the initial problem and IFM procedures could have subsequently created the connector potting problem. CORRECTIVE_ACTION: The potting problem has been corrected with an update to the manufacturing procedure to epoxy the Teflon cable to the connector and then pot. The voltages in the VIUs have been increased (approximately 0.2 volts) in order to handle the camcorder current surge. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 005:20:27	Problem	**FIAR** None	**IFA** STS-40-V-06	GFE
INCO-01, INCO-05	**GMT:** 162:09:52		**SPR** None	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: a. Text and Graphics System Hardcopier Jam Indicationb. Text and Graphics System Hardcopier Jam (GFE)

Summary: DISCUSSION: a. During the initial Text and Graphics System (TAGS) hardcopier uplink, and a number of time thereafter, a TAGS jam indication was annunciated. The indication was cleared by performing two page advances. (One advance is used to clear the indication and the other is used to cycle the Orbiter hard copier (OHC) to full image ready). This is indicative of a faulty sensor.

b. During the morning uplink on flight day 7, TAGS jammed. The crew worked through the appropriate malfunction procedures without success. TAGS was lost for the duration of the flight. CONCLUSION: TAGS was removed from OV-102 and sent to JSC for troubleshooting. a. The results from troubleshooting showed that paper sensor 3 was misaligned. This sensor is located at the entrance of the developer. The misalignment decreased the sensitivity of the sensor. b. The developer upper paper

guide condenses moisture which is a natural by-product of the developing process. Subsequent pages become stuck in this moisture which causes the paper to "accordion" and jam.   CORRECTIVE_ACTION: a. The sensor was fixed with a mechanical realignment. Should this anomaly recur in-flight page advances can be used to clear the faulty indication. b. All hardcopiers will be modified for STS-42 and subsequent missions. This modification includes a modified developer with no upper guide, which eliminates the sticking point and associated jams. Additionally, the developer exit is enlarged which will facilitate the clearing of any other jams.
EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 001:11:26	Problem	**FIAR**	**IFA** STS-40-V-07	OMS/RCS
PROP-02	**GMT:** 158:00:51		**SPR** 40RF06	**UA**	**Manager:**
			IPR 50V-0009	**PR**	
					Engineer:

Title: L5L Failed Off (ORB)

Summary: DISCUSSION: During STS-40, reaction control subsystem (RCS) vernier thruster L5L failed off due to low chamber pressure following a series of short-duration pulses. Redundancy management (RM) declared L5L failed-off when the chamber pressure reached only 18 psia when commanded to fire. As per Flight Rule 6-31, thruster L5L was reselected and hot-fired three times in an attempt to clear the contamination that could have caused the failure. Although the chamber pressure response improved with each pulse during the hot-fire and continued to improve throughout the remainder of the flight, the chamber pressure never achieved over 90 percent of its full operating value of 110 psia.

The thruster fail-off was most likely caused by iron-nitrate contamination in the oxidizer valve or trim orifice which significantly reduced oxidizer flow. Vehicle rate data during the hot-fire indicated that thruster L5L did not achieve full thrust (24 pounds). This data would indicate that contamination continued to affect oxidizer flow. Also, pressures recorded on the ground following the flight showed that the pressure transducer was biased low. Either of these, or a combination of the two most likely contributed to the low chamber pressures during subsequent firings.   CONCLUSION: The deselection of RCS thruster L5L was most likely caused by iron-nitrate contamination in the oxidizer valve or trim orifice. This contamination could significantly restrict oxidizer flow. Failure of the chamber pressure to reach its full operating value was most likely caused by contamination remaining in the oxidizer valve that resulted in an off-nominal mixture ratio or it may have been the result of the pressure transducer being biased low, or a combination of both. Investigation of this anomaly is continuing under CAR 40RF06.   CORRECTIVE_ACTION: KSC removed and replaced the RCS thruster and returned it to the vendor for failure analysis.   EFFECTS_ON_SUBSEQUENT_MISSIONS: None. A spare thruster was available on site.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 006:16:20	Problem	**FIAR**	**IFA** STS-40-V-08	PRSD,EPD&C
EGIL-01	**GMT:** 163:05:45		**SPR** 40RF07	**UA**	**Manager:**

IPR 50V-0005 **PR**

Engineer:

Title: Hydrogen Tank 3 Heater A Failed Off (ORB)

Summary: DISCUSSION: The cryogenic hydrogen tank 3 heater A failed off at 163:05:15 G.m.t. On-orbit troubleshooting verified that the heater was not functional in either the "auto" or "on" switch positions. The B system heater was still functional and it was used to deplete the remaining hydrogen from tank 3.

Troubleshooting at KSC found that the 5 Ampere fuse F6 in the cryogenics control box had opened. The fuse was replaced and the A system heater functioned normally. An abrasion was found in the wiring between the cryogenics control box and heater under a clamp that secured the wire to the structure. CONCLUSION: The abraded portion of the heater circuit apparently made contact with the cable clamp, and this resulted in a temporary short-to-ground which caused fuse F6 to open. CORRECTIVE_ACTION: The abraded portion of the heater circuit will be repaired. In addition, the wiring to this heater will be covered with convoluted tubing to protect against future abrasions under the cable clamp. All other vehicles will be inspected to determine if they also need convoluted tubing in this area. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 006:00:28	Problem	**FIAR**	**IFA** STS-40-V-09	TCS
MMACS-06	**GMT:** 162:13:53		**SPR** 40RF08	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: Loose Thermal Cover on Tunnel Adapter Top Hatch. (ORB)

Summary: DISCUSSION: While viewing payload bay camera video downlinked at approximately 162:13:53 G.m.t., a loose thermal cover was noticed on the tunnel adapter top hatch (the extravehicular activity hatch). A detailed review of earlier payload bay camera video showed that the hatch cover had been loose since early in the mission. A thermal analysis and evaluation of possible interference with nearby hardware determined that the loose cover would cause no impact to the mission.

The cover is held in place by five velcro loop straps. A review of the installation paperwork showed that the velcro straps had been verified to be properly attached after the lid/ring assembly had been installed on the airlock tunnel. A postflight inspection was performed by KSC personnel (Reference Corrective Action Record 40RF08-010). All five of the straps were found to be in the stowed (disengaged) position, indicating the cover had possible been reopened after installation and close-out. CONCLUSION: The cause of the loose thermal cover was improper configuration of the loop straps that secure the cover. The loose cover had no impact on the mission. CORRECTIVE_ACTION: KSC has changed an existing Thermal Control System (TCS) Test Preparation Sheet (TPS) to insure proper installation of the thermal cover by making the installation the last step of the mission kit work. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:**	Problem	**FIAR**	**IFA** STS-40-V-10	C&T
INCO-03	**GMT:**		**SPR** 40RF09 (LL), 40RF14 (LR)	**UA** 2-A0012	**Manager:**
				PR 2-A0027	
			IPR		**Engineer:**

Title: Lower Left and Lower Right S-Band Quad Antennas Communications Erratic (ORB)

Summary: DISCUSSION: Several unexplained forward link dropouts occurred throughout the STS-40 mission when using the lower left (LL) and/or the lower right (LR) S-band antennas.

CONCLUSION: Similar dropouts have occurred on previous Space Shuttle missions and are believed to have been caused by loose or burned coaxial cable connectors and intermittent bulkhead feedthrough connectors. Resolution of this problem has been deferred to the vehicle major modification effort, during which a radio frequency (RF) transmission path evaluation (which includes the coaxial cables and their connectors) will be performed for the S-band phase modulation (PM) system. This evaluation is intended to identify and correct any problems in the RF cabling which could cause S-band link dropouts. No postflight troubleshooting was conducted at KSC. CORRECTIVE_ACTION: The two lower quad antennas (LL and LR) have been removed and returned to the vendor for tests and repair. All PM system cabling will be demated, individually tested and then remated. The connectors will be individually torqued to specifications. Any damaged or failed cables will be replaced with spares. The four 576 bulkhead feedthrough connector (one in each transmission path) will also be removed and replaced with verified spares. Upon completion of the coaxial cable evaluation and reinstallation of the quad antennas, a system level retest will be performed. EFFECTS_ON_SUBSEQUENT_MISSIONS: None

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** Postlanding	Problem	**FIAR**	**IFA** STS-40-V-11	TPS,STR
None	**GMT:** Postlanding		**SPR** 40RF10	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: Right-Hand External Tank Door Thermal Damage. (ORB)

Summary: DISCUSSION: During the postflight-inspection at Edwards Air Force Base, it was noted that the right-hand External Tank (ET) door forward centerline latch fitting exhibited melting and erosion. Adjacent tiles exhibited slumping. The forward end of the latch point was eroded along the forward 2 inches approximately 0.1 inch in depth with the outer corner on the forward tip being melted away. The melted material fused over the edge of the adjacent tile (395055-046). Since there is no reference point to use for a measurement and the left-hand door has a different configuration, it is not possible to determine how much of the -046 tile just forward of the latch fitting was slumped. A flow path was identified from the point of the thermal damage (the forward outboard corner) along the inside of the door between the thermal barrier and the internal pressure seal (bulb seal) which followed the wraparound bracket to a void (structural clearance between the bracket and aft fuselage structure) located at the aft outboard corner of the umbilical cavity which opened into the aft compartment. The edge of the aft tile (no. 395055-045) adjacent to the latch fitting may have experienced minor thermal damage. Evidence was found that metal from the molten latch tip had spewed backwards along the flow path and deposited on the tile. the flow path was isolated to the area outside the pressure seal. The internal pressure seal and thermal barrier were intact with no evidence of damage or severe overheating. The Kapton film located inside the cavity between the pressure seal and thermal barrier was inspected and found to be in good condition with no evidence of overheating; therefore, the air and molten latch patch (thermal barrier) was intact with typical outer mold line (OML) heat discoloration.

The door closure was verified and the step-and-gap data were taken to analyze the door-to-structure forward facing step for possible localized overheating conditions. The external step-and-gap data were all within specifications, however, a disparity in the latch point to adjacent inconel finger step was identified. This created a small void between the latch point and the thermal barrier latch patch which coupled with the structural clearance opening in the aft of the umbilical cavity caused a plasma flow to ingest into the umbilical cavity and aft compartment causing thermal damage to the door. No damage to any other compartment was observed or recorded. The same structural clearance between the wraparound bracket and aft fuselage structure was found on the left side of the vehicle. Both clearances were inspected on all vehicles and filled with room-temperature vulcanizing material (RTV) per the design requirement. CONCLUSION: The thermal damage was caused by a unique combination of factors and is not a safety-of-flight issue. The flow path was isolated to the area outside the internal pressure seal, and the flow cooled immediately upon entering the umbilical cavity. CORRECTIVE_ACTION: The OML for this area has been verified on the other two vehicles. A small forward face step has also been identified on OV-104 which will allow protruding latch point metal into the air stream, however, the flow path has been eliminated and there is no issue for STS-43. The structural clearances between the wraparound bracket and aft fuselage structure on all vehicles have been inspected and filled with RTV per the design requirement. Proper door closure has been verified on this vehicle, and the latch fitting, discrepant inconel finger and two adjacent tiles will be removed and replaced.
EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 009:01:13	Problem	**FIAR**	**IFA** STS-40-V-12	APU
MMACS-07	**GMT:** 165:14:38		**SPR** 40RF11	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: APU 1 Fuel Test Line Temperature Rise (ORB)

Summary: DISCUSSION: Shortly after the start of auxiliary power unit (APU) 1 for entry, APU 1 fuel test line temperature 1 (V46T0184A) rapidly rose from approximately 87 ?F to a value above the fault detection and annunciation (FDA) limit of 95 ?F. The temperature peaked at 99 ?F before beginning to decline. After an

"SM0 THRM APU" caution and warning message was received at 95 ?F, the crew repositioned the Tank/Fuel Line/H2O System 1B heater switch on panel A12 from AUTO to OFF. Heater switch 1A remained OFF. A review of the data, however, shows that the test line temperature had peaked and was declining when the 1B heater switch repositioning was accomplished.

Similar sudden spikes on APU fuel test line temperatures have been seen during APU operation on different APU's and Orbiters. A temperature rise varying from 4 to 16 ?F has been observed. If this temperature rise coincides with the top of heater cycling (which can reach 91 ?F), an FDA violation may occur. On STS-28 (OV-102), a 12 ?F rise near the upper limit of heater cycling at 88 ?F resulted in violating the FDA. On STS-40, the 87 ?F test line temperature at initiation of the rise was also near the top of the heater cycle. The APU fuel test line is a closed line which T's off of the main fuel line. The heater element that is wrapped around the test line produces localized warm spots in the fuel within the line. After APU start, fluid movement within the closed test line causes these warm slugs of fuel to pass by the temperature sensor locations, thereby producing a spike on the temperature sensor output. CONCLUSION: The sudden rise in the APU 1 fuel test line temperature 1 was a fluid heating phenomenon that has been seen on prior flights. It was most probably caused by a warm slug of fuel within the fuel test line. This condition appears to be an operating characteristic of the system which does not affect APU operation. CORRECTIVE_ACTION: Since the current limit of 95 ?F was set to aid in determining a failed-on heater in the cold prelaunch environment, revising the entry FDA limit upward to avoid the occasional nuisance alarm is being considered. Thermal analysis is in work to determine how raising the FDA limit could affect the crew response time in the event of a failed-on heater. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 005:08:35	Problem	**FIAR**	**IFA** STS-40-V-13	GFE
INCO-04, INCO-06	**GMT:** 161:22:00		**SPR** None	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: A. Loss of Communication Audio Interface Unit-DB. Loss of Communication Audio Interface Unit-E (GFE)

Summary: DISCUSSION: A. The crew reported the loss of communications on audio interface unit (AIU)-D which was installed in Spacelab. The unit was never recovered and remained unusable for the duration of the mission.

B. The crew reported a total loss of audio communications to both crew remote units (CRU) on AIU-E installed in Spacelab. In-flight troubleshooting isolated the problem to AIU-E. The audio loss was temporary, lasting approximately 1.5 hours. AIU-E operated nominally for the duration of the mission. CONCLUSION: A. Postflight troubleshooting showed a blown fuse in the intercommunications master station. Once this fuse was replaced, AIU-D operated correctly. The most probable cause of the blown fuse was connecting and disconnecting the headsets without turning the power OFF. Marshall Space Flight Center closed this problem as "unexplained". B. Postflight troubleshooting could not recreate the in-flight anomaly. Troubleshooting will continue when the units is returned to JSC. CORRECTIVE_ACTION: End-to-end functional tests were performed on both of these units at KSC. Troubleshooting will continue at JSC once the units are delivered. Functional units have been installed in the Spacelab module to support STS-42 (OV-103). EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Should these anomalies recur in-flight, Spacelab crew members can change channels on the CRU's to operate with the Orbiter AIU's. Handheld microphones are also available.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** Postlanding	Problem	**FIAR**	**IFA** STS-40-V-14	GFE
None	**GMT:** Postlanding		**SPR** None	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: A. 16 mm ET Umbilical Camera AnomalyB. 35 mm ET Umbilical Camera Anomaly (GFE)

Summary: DISCUSSION: A. Postflight film processing revealed three separate anomalies involving the 16-mm camera with the 5-mm lens. These were:

1. The film usage was less than expected. 2. None of the exposed film contained an image. 3. After processing at JSC, approximately 30 feet of unprocessed film was found on the take-up reel. The 16-mm camera operates at 240 frames per second and will achieve the full operating speed within 3/4 of a second from startup. The camera is configured for an "easy start" of the motor and to preserve this configuration, the camera is not operated from the time it is loaded until it is used in flight. The camera runs for 10 seconds at Solid Rocket Booster (SRB) separation and again for 70 seconds at External Tank (ET) separation. The camera is started and stopped electronically for the SRB separation run and for the ET separation run it is started electronically but stopped by a timer. Prior to sending the film to the Johnson Space Center (JSC) film laboratory for processing, any remaining film on the supply reel is processed through the camera onto the take-up reel. The camera is usually operated for 40 to 45 seconds to assure that all the film is on the take-up reel. Once this process was completed and the camera opened, it was discovered that there was still film on the supply reel indicating that less film was used than expected. There were not any problems encountered in the laboratory in operating the camera postflight before the anomalies were discovered. After the film had been processed and removed from the developer, it was noticed that approximately 30 feet of film remained to be processed because of a break in the film. It was noted also that there was sprocket damage to the film and several feet of film had been folded accordion style. About 30 feet of film is used to load the camera, which indicated that the break in the film apparently occurred as the camera started up for the SRB separation run. Analysis of a damaged portion of the film by Eastman Kodak indicated that the camera was jammed by the film. This is consistent with the accordion-type folding of the film found after the break. The film used on this flight was manufactured in 1984 and is thinner than the film normally used in the rest of the program. The use of the thinner film is required so that the desired amount of film can be loaded onto the limited space available on the supply reel. The analysis by Eastman Kodak indicated that the age of the film along with the dry nitrogen purge preflight caused the film to become brittle. Postflight, the failed camera was loaded to a full roll of film and operated properly in the laboratory. B. During the postlanding inspection, the film in the 35-mm ET umbilical camera was found to have sprocket damage and multiple exposures on the first frame. The troubleshooting performed on the camera revealed that the back cover was either bent or warped. This bending of the cover displaced the pressure plate springs on the stops. In addition, it was noticed that the film cassette had an area that was rubbed to a shine by the fork used to pull the film from the canister. The inspection revealed that the film canister spool was bent. The camera had no external marks that would indicated that is was dropped or how it became damaged. CONCLUSION: A. The most probable cause of

the film breakage in the 16-mm camera is that the film became slack due to launch-induced vibrations. This slack allowed the film to move at a slower rate of speed than the sprockets which in turn caused the observed damage to the film and ultimately jammed the camera. The brittleness of the film was a contributing factor in the film break. The jam was probably cleared by landing loads, and this permitted the camera to operate normally in the laboratory. The film jam is also the reason there was not an image on the film that was thought to have been processed by the camera, and also accounts for the less-than-expected film usage. B. The most probable cause of the 35-mm camera anomaly is that the pressure placed on the film by the pressure plate caused the film to beak. The bent or warped back cover caused the pressure plate to exert the excessive pressure on the film. CORRECTIVE_ACTION: A. The camera was tested with flight film and no anomalies were found. The camera has been returned to flight status. The existing supply of film will be used until exhausted. B. The camera has been returned to the vendor for repair.
EFFECTS_ON_SUBSEQUENT_MISSIONS: A. Due to the age of the film and the operational environment of these cameras, it can be expected on occasion to experience film breakages of this nature on future flights. B. None.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** 000:09:00	Problem	**FIAR**	**IFA** STS-40-V-15	INST
INCO-07	**GMT:** 156:22:25		**SPR** 40RF12	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: Ground Command Interface Logic Payload Data Interleaver Command Anomaly (ORB)

Summary: DISCUSSION: When the command was uplinked through the ground command interface logic (GCIL) to turn OFF the payload data interleaver (PDI) at 156:22:25 G.m.t., the PDI switch-scan parameter (MSID V75S5100E) changed state (ON to OFF) reflecting the GCIL command instead of the actual switch position. The crew toggled the switch and the parameter indicated correctly.

CONCLUSION: Postflight investigation explained the anomaly as reverse current feedback through the GCIL driver. The possibility of a reverse feedback ("sneak circuit") condition in the GCIL where current is backfed from the PNL (panel) drivers through the CMD (command) drivers was known to exist. This was identified at Palmdale in 1983 during OV-102 pre-acceptance tests. Although the STS-40 anomaly scenario differs from the 1983 scenario, the cause of both anomalies is the same. The 1983 occurrence effected a criticality 1 function. At that time, criticality 1 functions effected by reverse feedback through GCIL drivers were identified on all vehicles and protected with the use of blocking diodes. The vehicles were accepted with this known deficiency for criticality 2 and 3 functions. CORRECTIVE_ACTION: Rockwell and Smith Industries have been tasked to perform a detailed analysis to define the GCIL controls which could have feedback and the effects resulting from the feedback. The deadline for the completion of this effort is 12-20-91. EFFECTS_ON_SUBSEQUENT_MISSIONS: None. Should this anomaly recur, the correct measurement reading can be obtained by placing the GCIL switch in the PNL position.

Tracking No	Time	Classification	Documentation		Subsystem
MER - 0	**MET:** Postlanding	Problem	**FIAR**	**IFA** STS-40-V-16	MPS
None	**GMT:** Postlanding		**SPR** 40RF13	**UA**	**Manager:**
			IPR None	**PR**	
					Engineer:

Title: LH2 Umbilical Guide Pin Bushing Dislodged (ORB)

Summary: DISCUSSION: A postflight review of film from the 16 mm camera located in the LH2 ET umbilical well, revealed a cylindrical object drifting past the camera and moving away from the Orbiter. The object entered the field of view approximately 44 seconds after separation and was visible for about 2.5 seconds. The object appeared to be metallic and hollow with an approximate length/diameter ratio of 1.2. Inspection of the OV-102 Orbiter umbilicals after landing showed no missing hardware. Further review of the ET separation sequence films indicated that the inconel-718 bushing was apparently missing from the outboard LH2 ET umbilical guide pin hole. This determination was based on the lack of sheen around the bushing hole (i.e. the shiny bushing was missing) and the calculated hole diameter at the bushing site was neared to the inner diameter of the bore rather than the inner diameter of the bushing. The bushing sheen was visible at the LH2 inboard and both LO2 ET bushing locations.

Each ET umbilical has two guide pin bushings. Their purpose is to accept the guide pins located on the Orbiter umbilical that provide alignment control during umbilical mating as well as during umbilical separation after MECO. The bushing has a stepped outer diameter of 1.750 and 1.648 inches, an inner diameter of 1.250 inches, and a length of 1.432 inches. The bushing is designed to remain within the ET during separation. The main concern with this bushing becoming dislodged during separation is that it may migrate into the Orbiter's ET umbilical door closure mechanism, causing a failure of the door to close. However, this is unlikely since the relative motion of the debris is away from the Orbiter and several translation maneuvers are performed prior to umbilical door closure that tend to further move the vehicle away from the debris. Therefore, the likelihood of the bushing finding its way into critical parts of the mechanism is small; and if it did, the umbilical door could be recycled in an attempt to remove the bushing. An investigation into why the bushing became dislodged has, thus far, been unable to determine the reason. These bushings are retained in the body by a shrink fit interference that by blueprint is between 0.0005 to 0.0015 inch at ambient temperatures. This interference increases by an additional 0.0027 inch at LH2 temperature because of differential contraction of the aluminum body/inconel-718 bushing. The combined minimum interference should require an axial load of about 8100 pounds to dislodge the bushing. The mating Orbiter pin has two diameters that provide a minimum diametrical clearance of 0.011 inch for the first 0.36 inch of separation travel, then 0.014 inch of diametrical clearance for the remaining 1.2 inches of pin-to-bushing travel. This allows for up to 3 degrees of "angular" misalignment. A review of the build documentation on this disconnect failed to provide an explanation into the apparent missing bushing. Recorded measurements of the bore actual diameter showed it provided an interference of 0.0004 inch above the minimum required. Bushing dimensions were not recorded, but it was from a lot where all dimensions were noted as being within requirements. This indicates that the dislodging forces should have been in the 8000 pounds range. ET separation dynamics were nominal. A postflight review of umbilical plate retractors showed no evidence of binding or cocking of the umbilical plate. An inspection of the OV-102 pins revealed a number of light burnish marks on the outboard guide pin. However, all burnish marks were visible on preflight photographs except for a linear burnish mark noted at the

11:00 position (12:00 being toward the nose) and extending from the beginning of the tapered section to the end of the pin. A similar mark was also noted on the LH2 inboard guide pin, as well as on all four pins on OV-103. A review of prior OV-102 ET separation photographs suggests that the same location may be missing its bushing on STS-5 and STS-28. The film of STS-3 shows that the bushing is definitely in place. Other mission films are either too dark and/or the location is too obscured by H2 ice to determine the bushing status. STS-40 was the second flight for this umbilical (installed from OV-105 following the STS-35 mission scrub for LH2 leakage), but first flight film is not available because STS-35 separation occurred in darkness. Possible causes of the bushing removal include a manufacturing and/or inspection problem that results in improper bushing-to-ET umbilical plate fit or a cracked ET umbilical plate around the bushing that reduced the bushing-to-body interference fit. Also, excessive side and/or angular forces (from an undetermined source) could result in high axial friction forces that could pull the bushing during separation and hydraulic retraction of the Orbiter umbilical plate. CONCLUSION: The LH2 outboard ET umbilical guide pin bushing became dislodged from the ET umbilical plate during ET/Orbiter separation due to unknown reasons. The bushing translated away from the Orbiter and had no effect on the mission. CORRECTIVE_ACTION: For STS-43, the bushings were verified during mating to be at least finger tight. Several inspection techniques for ET bushing installations have been discussed, but the issue is not as yet settled. Additionally, several tests are being considered to determine the actual pin bushing side load required to dislodge a bushing. Also, testing is being considered to define the distinguishing marks associated with forces necessary to dislodge the bushing. EFFECTS_ON_SUBSEQUENT_MISSIONS: None.